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PLANETARY AND ASTEROID MISSIONS -- GETTING THERE

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ANCHORING SPACECRAFT TO ASTEROIDS

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Abstract

In this hardware project, the students developed ideas for attaching objects to the surface of small moons or asteroids. A device was designed, and built in the university machine shop, that uses a projectile shot into concrete, thereby attaching a model spacecraft to the landing site.

Introduction

Past exploration of the solar system has been concerned almost exclusively with large objects such as planets, the sun and our moon. However, increasing attention is now being paid to the smaller bodies, not only because of the wealth of information they may contain, but also because of their potential as propellant sources, support points for long space missions and bases of operations. Their usefulness in these areas is due in large part to their extremely low gravity which makes them very accessible and allows easy departure.

The microgravity encountered on these bodies does present some challenges, however. Not only is it easy for equipment to shift orientation or leave the surface due to internal movements, but for any action such as core-boring or drilling, a positive restraint is needed. This project seeks to address the problem of keeping probes, landers, and equipment on the surface of these tiny bodies.

Anchoring Concepts

Several ideas for anchoring devices with various capabilities were studied. Initial ideas fell into the categories of tethered projectiles, drills and hot probes. The hot probes melt into the surface, then cool down to fuse with it. The question of a deep regolith inspired ideas such as a combination of a large auger which would work its way down to the rock and then fire a tethered projectile contained in its tip. Another idea wass a "mole" machine to burrow down through the regolith letting out a cord as it went (Fig. 1.1). When it reached rock, the "mole" would fuse itself to the rock or fire a projectile. Various designs allowed multiple anchor placement without moving the lander. These systems increase complexity, but improve anchoring reliability.

The tethered projectiles appeared to be the best choice in terms of reliability, simplicity, cost, weight, and mission flexibility. Design of the actual penetrator would be a major study in itself. Penetrators which use superheated liquid in their tip may greatly reduce velocity requirements. Recoilless penetrator systems may offer less weight and greatly decrease shock loads to the lander, but flexibility is more limited as exhaust gases must not impinge on sensitive components.

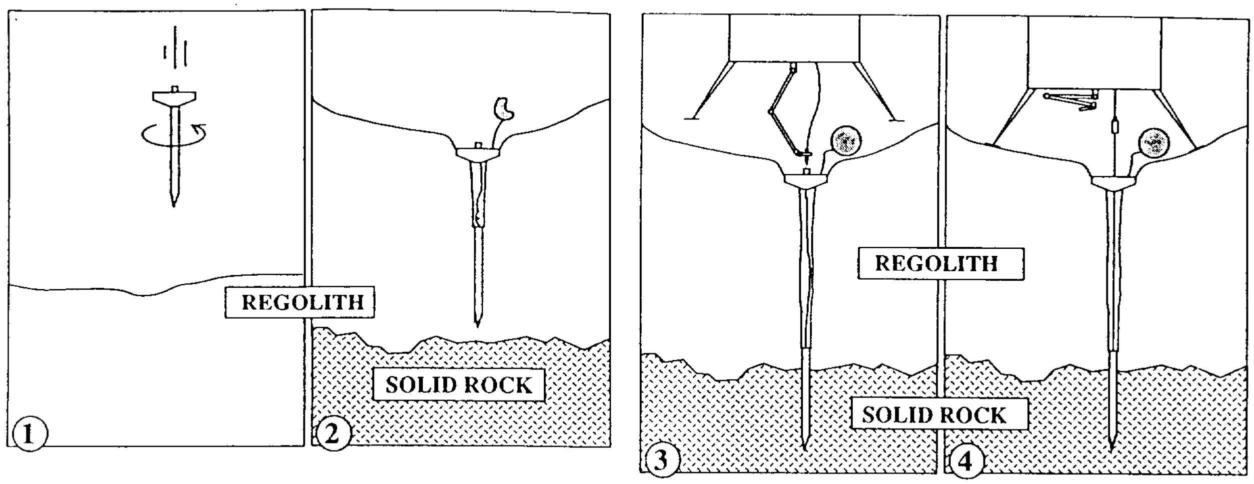

Figure 1.1: One of the various concepts for anchoring systems.

Mechanical and Electrical Design

The design solution (Figs. 1.2a,b) is based largely upon a common concrete nail gun. It employs a hammer, guided by a tube and set in motion by a spring. The hammer strikes a cartridge which ignites and fires a projectile down the barrel into the surface. A motor winds up the line, which is attached to the projectile prior to insertion into the barrel. This pulls the lander to the surface and holds it there. Other major components of the design include assemblies for lowering the barrel to the surface and cocking the hammer, both of which employ a power screw driven by a motor. The trigger mechanism is operated by a solenoid.

As the device was intended to be used as a demonstrator, several support systems had to be designed and built including a tripod lander. A system of long springs was employed to simulate a weightless environment. Other systems include the safety cage, control electronics, and various simulated asteroid surfaces.

Possible Uses on Phobos

Phobos was studied as a possible near term use for this type of device. Phobos is a small moon of Mars which is receiving attention as a possible propellant source and way station for a manned exploration of Mars. Various characteristics such as composition, topology, regolith and gravity characteristics were investigated. The need to anchor equipment is evident as the gravity is roughly 1/2000th that of earth. Although most of Phobos consists of solid rock (probably carbonaceous chondrite), the presence of a regolith of variable depth (possibly up to 300 meters) raises questions about the ability of the device to work on all points of the surface. There is evidence that the regolith is very thin near recent impact areas, the tops of ridges and the rims of craters.

Although there were many ideas for use in outer space, few fit the safety requirements for a laboratory demonstrator. The concept chosen for demonstration was the simplest and employed common technologies which were already well proven.

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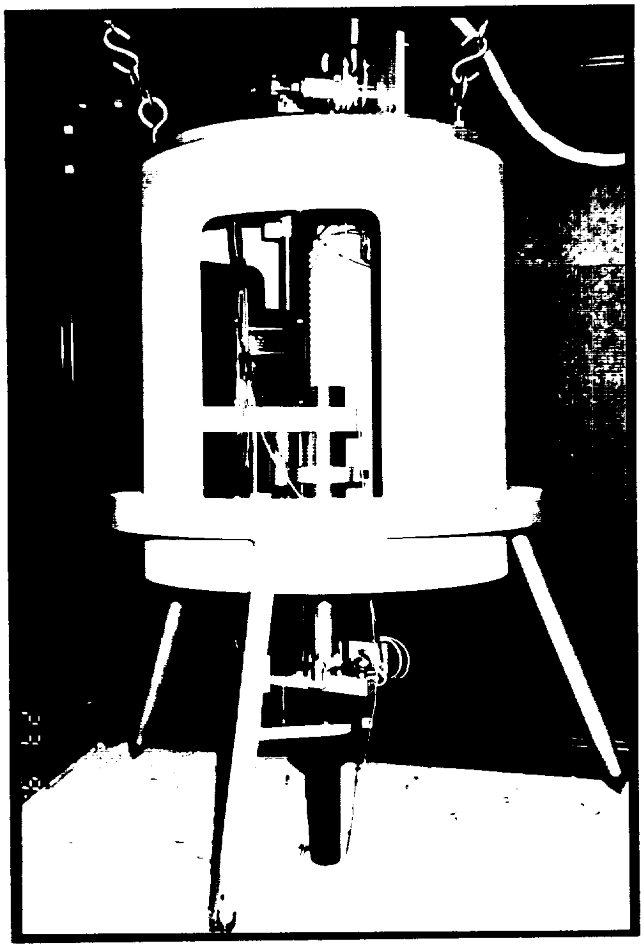


Figure 1.2a: Student-designed anchoring device

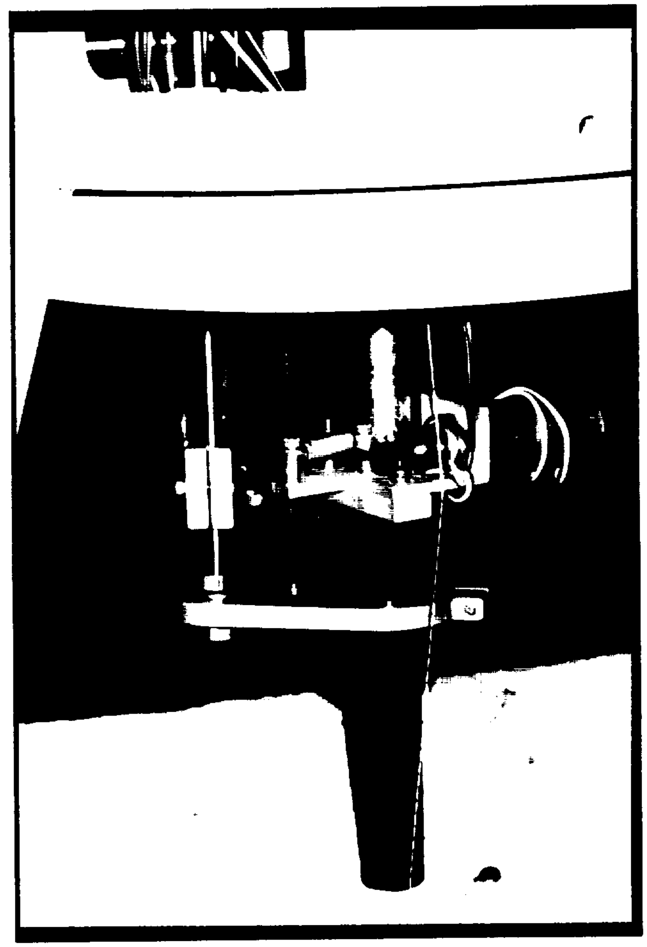


Figure 1.2b: Close-up of the barrel. The hammer has been uncocked.

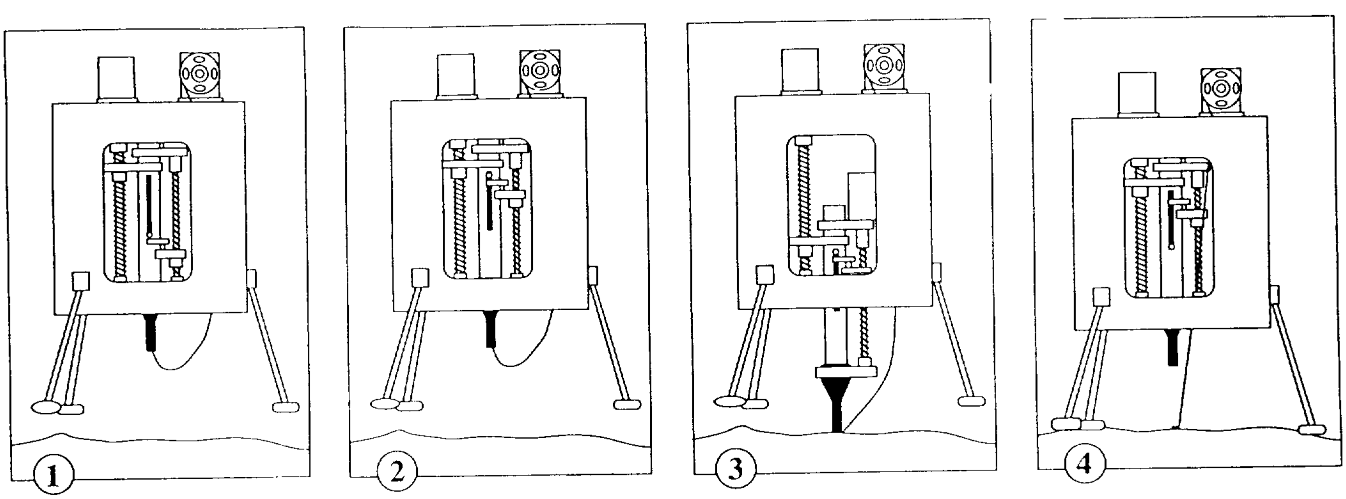


Figure 1.3: Operation of the final design. The spacecraft arrives at the surface (1). The hammer is cocked (2). The barrel is lowered and fired (3). The gun is raised as the spacecraft is lowered to the surface by winding the cord (4).

Conclusions

Testing has shown the device to work on concrete surfaces of various compositions. The device also works on ice if the projectile is heated prior to insertion into the barrel. Tests with ice and weaker concrete have suggested that longer projectiles would work better in these surfaces, but for reasons of laboratory safety the barrel was designed only for small nails. Only the lightest powder loads are used in the demonstrator. The construction industry uses loads up to seven times higher in nail guns of about the same size as our device. This implies that a device no larger than the demonstrator, without the housing, may be more than capable of performing the necessary anchoring on real missions. A device designed for electrically fired cartridges and lower reusability could be very lightweight and compact. Due to the variable surface characteristics of Phobos it is possible that no single device will work in all situations.

ION ENGINE PROPELLED EARTH-MARS CYCLER

Daniel Limonadi - Teaching Assistant

Abstract

The goal of this project was to perform a preliminary design of a long term (life span ≥ 15 yrs), reusable transportation system between Earth and Mars which would be capable of providing both artificial gravity and shelter from solar flare radiation. The heart of this system is assumed to be an ion engine propelled cycler spacecraft launched several years in advance of a manned mission. Several Mars transportation system architectures and their respective space vehicles were designed.

Introduction

General interest within the space community regarding Mars transportation system architectures has been on the rise since the mid 1980's. One class of these systems seeks to provide artificial gravity and shelter from radiation storms caused by solar flares during transit to and from Mars. Due to the expected structural constraints and large mass that an interplanetary transit vehicle meeting the above requirements is expected to have, low thrust, but very high specific impulse ion engines are assumed to power the vehicle (henceforth called the Cycler). Several aspects of such a transportation system were investigated. They included: 1) preliminary thermal, power, propulsion, structural design, and transit time requirements of a cycler vehicle and its particular orbital trajectory; and 2) the requirements placed on the planetary vehicles which would transport crew and cargo between interplanetary transit vehicles (the cyclers in the case of personnel transfer) and the surface of Mars and/or orbiting platforms. Emphasis was on trajectory design, the propulsion system and aerobraking.

Assumptions and Requirements

Several baseline assumptions were made which where incorporated into the various vehicle designs:

- propulsion system baseline performance ("Cycler")

- ion drive; Isp = 10000 sec.
- LOX/LH2 system used on planetary vehicles ("Taxis"); Isp = 480 sec. based on a mixture ratio of 7:1

- Nuclear reactor power to mass ratio; 8 kg/kW

•*Mars Infrastructure:* Mars infrastructure assumptions played a crucial role in defining the vehicle requirements and orbits used near the planets. The infrastructure varied from design group to design group, though a basic item which all groups included was some form of refueling station at Mars, either at Phobos or in a low orbit. Other options generally included LOX production on the surface of Mars, based on ideas presented by Zubrin[1].

•*Earth Infrastructure:* The Earth infra-structure was assumed to be in place starting at the time of the cycler vehicle assembly and generally included a space station in LEO, a vehicle assembly and servicing facility, a Moon base which supplied LOX to the transportation system, and orbital transfer vehicles (OTV's) to service the Earth-Moon system.

To help constrain portions of the vehicle design, a set of baseline requirements were made:

•The largest pressure vessel and other fabrication intensive components of any vehicle were required to be able to be placed into orbit by a launch vehicle currently in use or on the drawing boards; e.g. Shuttle, Spacelifter, Energia.

•The distance between the center of mass and crew quarters of the cycler vehicle was to be such that the Coriollis acceleration associated with a spin rate sufficient to induce 0.5g artificial gravity in the crew compartment would not be greater than 5% of the induced artificial gravitational acceleration, based on a assumed walking speed of 0.5 m/s relative to the vehicle.

Orbit and Trajectory Design

The orbits and trajectories analyzed for the Mars transportation system fall into two broad groupings: 1) interplanetary and 2) planetary (much like the vehicles that use them).

Interplanetary Orbits

One of the most important decisions to be made by the design groups was the type of interplanetary transfer orbits for the cycler spacecraft. Three classes of orbits where investigated: conjunction class minimum energy orbits; VISIT orbits; and Up/Down Escalator orbits. Table 1 shows some constraining parameters associated with these orbits. The basic trade-off between the choices focused on ΔV requirements for planetary access, frequency and regularity of cycler encounters, and crew transit time between cycler and destination planet. The final choice of the cycler orbit strongly affects both cycler spacecraft and planetary transfer vehicle design. Planetary approach velocities and transfer times especially play a defining role in design requirements for the crew/cargo planetary transfer vehicles.

Table 1: Cycler orbit constraining parameter comparison[2]

Parameters	**Visit - 1**	**Visit - 2**	**Up escalator**	**Down Escalator**	**Conjunction[3]**
Frequency of Earth encounters [yr.'s]	5.0	3.0	2.14	2.14	
Frequency of Mars Encounters [yr.'s]	3.75	7.5	2.14	2.14	
Earth to Mars flight time [yr.'s]	0.5-3.0	1.0-2.4	.43	1.71	0.7
Mars to Earth Flight Time [yr.'s]	0.7-3.3	0.6-2.1	1.71	.43	0.7+
Earth Encounter V [km/s]	4.2-4.8	3.7-4.0	5.7-6.2	5.4-6.0	2.29-3.51
Mars Encounter V [km/s]	3.7-4.1	2.6-2.8	6.1-11.7	6.6-11.6	1.98-3.28
Earth Encounter Distance [R_E]	6.9-SOI	8.3-SOI	1.2-1.9	1.2-1.8	
Mars Encounter Distance [R_M]	1.5-40.7	2.0-18.5	1.3-29.1	1.3-9.4	
Midcourse adjustment, 15 years [km/s]	0	0	1.7	2.0	
Max. Earth access V, 14 days [km/s]*	5.5	5.2	4.8	4.7	
Max. Mars access V, 14 days [km/s]*	2.9	2.2	9.4	9.2	

* Sum of ideal injection and rendezvous maneuvers. SOI : Sphere of influence of given body.

The ΔV requirements required of a cycler spacecraft in conjunction orbits make it impossible to rely solely on high efficiency ion propulsion systems since these systems are inherently low thrust. Thus the main candidate orbits for cycling spacecraft are the VISIT and Up/ Down Escalator orbits. However, the conjunction orbits do represent the minimum energy one way trajectory and are thus ideally suited for cargo transfer vehicles which are not restrained by requirements protecting human cargo.

Planetary Trajectories

The type of planetary trajectories executed by vehicles of the Mars transportation system are not strongly affected by the choice of interplanetary transfer orbits. However, the magnitude of the associated ΠV's for each maneuver is strongly dependent on the type of cycler orbit chosen. Table 2 shows a sample list of trajectory sequences and associated ΔV requirements for shuttle vehicles operating between cycler spacecraft, target planet's parking orbit, and planet surface. The values given in Table 2 are based on worst case Up/Down escalator orbit encounters at Earth and Mars. Insertion into a 3600 km parking orbit is assumed at Mars, and a 6671 km parking orbit at Earth (distance taken from center of planet).

Table 2: Sample Trajectory Sequences at Earth and Mars for Planetary Transfer Vehicles (Calculations are based on worst case Earth and Mars encounter conditions for Up/Down Escalator orbits. Three days are allowed for shuttle transfer to Mars, and 1 day for transfer to the Cycler at Earth.)

		ΔV [km/s]
1.	Hyperbolic Earth Escape	4.69
2.	Cycler Rendezvous (Earth)	0.07
3.	Transfer to Mars intercept	0.372
4.	Retrofire	1.77
5.	Mars aerocapture	6.39
6.	Orbit circularization	0.02
7.	Deorbit	0.22
8.	Ascent to parking orbit	4.4
9.	Hyperbolic Mars Escape	9.13
10.	Cycler Rendezvous (Mars)	0.10
11.	Transfer to Earth Intercept	0.07
12.	Earth aerocapture	5.62

The most significant differences in planetary ΔV requirements arising from cycler orbit choices are found in the hyperbolic escape requirement at Mars and the ΔV required for transfer between the hyperbolic trajectories of the cycler vehicle and the hyperbolic trajectory required by the shuttle to intercept the target planet or return to the cycler from a planet. A simple formula relating the ΔV cost of changing these hyperbolic trajectories is given by Friedlander[3] et al[1], $\Delta V = \Delta B/\Delta T$, where ΔB is the difference in the impact parameter of the two trajectories and T is the time allowed from injection to rendezvous. Due to the large approach distances of the cycler spacecraft traveling in Visit orbits (beyond the sphere of influence of the Earth in the worst case) relatively large ΔV penalties are required to keep the crew transfer times reasonably short i.e. less than 21 days. In the case of the Up/Down escalator orbits the approach distance is short enough to enable transfer times as short as 1 day with little penalty. The main concerns regarding transfer times in these situations deal with minimum shuttle size to keep the crew relatively comfortable for the duration of the transfer, and more importantly, minimize the risk to crew members of exposure to radiation caused by solar flares. Although the Up/Down Escalator orbits have the advantage of a small ΔV penalty for short crew transfer time, the very large ΔV requirements for injection into hyperbolic rendezvous with the approaching cycler spacecraft at Earth, and especially Mars, place a great demand on propulsion requirements. Unless it is required that crew transfer times for the VISIT orbit cases be reduced to much shorter time periods than one or two weeks the Up/Down Escalator orbits require more propellant.

Vehicle Design

The Vehicles comprising the Mars Transportation System can be grouped into two major categories: planetary vehicles and interplanetary vehicles. This categorization is based on where the particular vehicle carries out its principal mission. The individual vehicles and their primary roles are presented below:

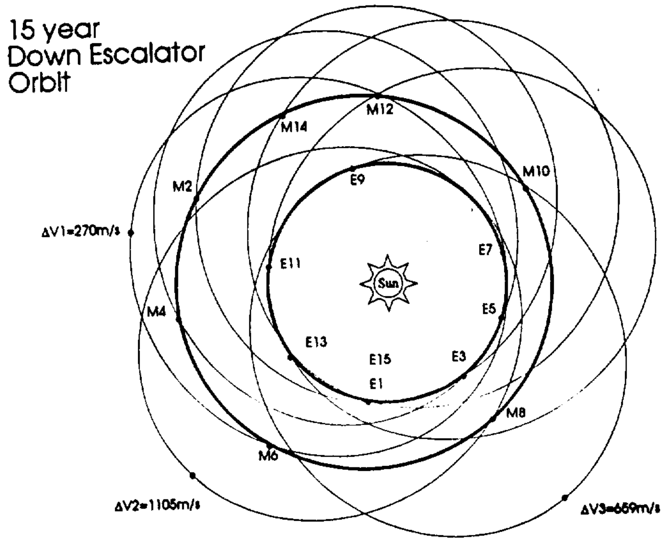


Figure 2.1: Up/Down Escalator Orbit Evoluation over a 15 Year Period. Planet Encounteres number in Sequence or Occurrance; *E* - Earth, *M* - Mars Encounter. (Scale 1.5 in = 1 AU).

Inter-Planetary Vehicles

Elements of the Mars Transportation system operating primarily in interplanetary space include the cycler space craft and cargo delivery vehicles which use minimum energy orbits.

Cycler Spacecraft.

The cycler spacecraft provide transportation between Earth and Mars for all personnel. Their size and trajectory choice are the primary factors which differentiate these vehicles from other proposed Mars transfer vehicles. As mentioned in the Introduction, the primary requirements for the cycler vehicles are an artificial gravity environment and radiation storm shelters. In addition, there is need for tanks to contain the argon fuel for the ion drives, docking and possible hangar facilities for Taxi spacecraft, antennae, a nuclear reactor and associated shielding, radiators and power conversion plant, and, depending on the particular design solar arrays, greenhouses, and remote sensing equipment. Figures 2.2 and 2.3 show some Taxi, OTV, and cycler vehicle design layouts. Table 3 provides mass estimates corresponding to the cycler vehicle in Fig. 2.3a.

Cargo Delivery Vehicles.

These vehicles are necessary to provide the ability to move large amounts of cargo between Earth and Mars. They basically consist of cargo pallets fitted into an aeroshell and provided with a propulsion system whose sole purpose is to provide attitude control, orbital correction ΔV's to insure proper atmospheric entry geometry at Mars, and to deliver the cargo into a circular parking orbit upon completion of the aerobrake maneuvers at Mars. Some initial mission designs had the large cargo delivery carried out by cycler spacecraft in an attempt to minimize vehicle types. Upon further analysis it was deemed that the inefficient use of propellant due to the ΔV injection requirements into higher energy Up/Down Escalator and VISIT orbits vs. those required for low energy conjunction orbits outweighed the benefits a of smaller infra- structure.

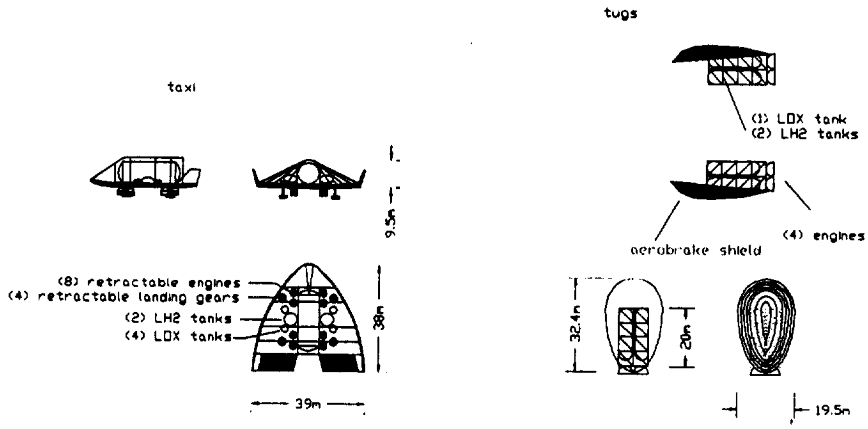


Figure 2.2: Left - Taxi Vehicle Utilizing High L/D Body Configuration. Right - Orbital Transfer Vehicle with Raked-Off Elliptical Cone Aerobrake.

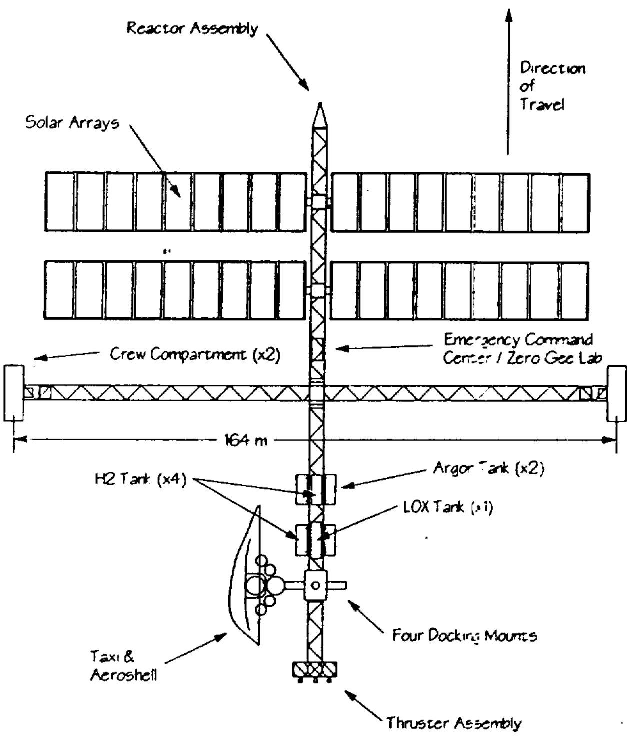


Figure 2.3a: Typical Cycler vehicle layout. Central truss structure is de-spun. Solar arrays are mounted to allow rotation about central truss and primary array axis to ensure maximum exposed surface area.

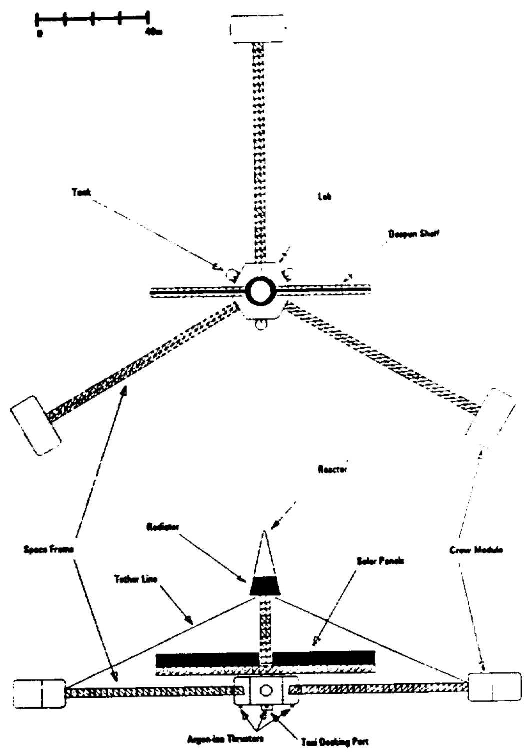


Figure 2.3b: Cycler design with vehicle spin axis equal to axis of maximum moment of inertia.

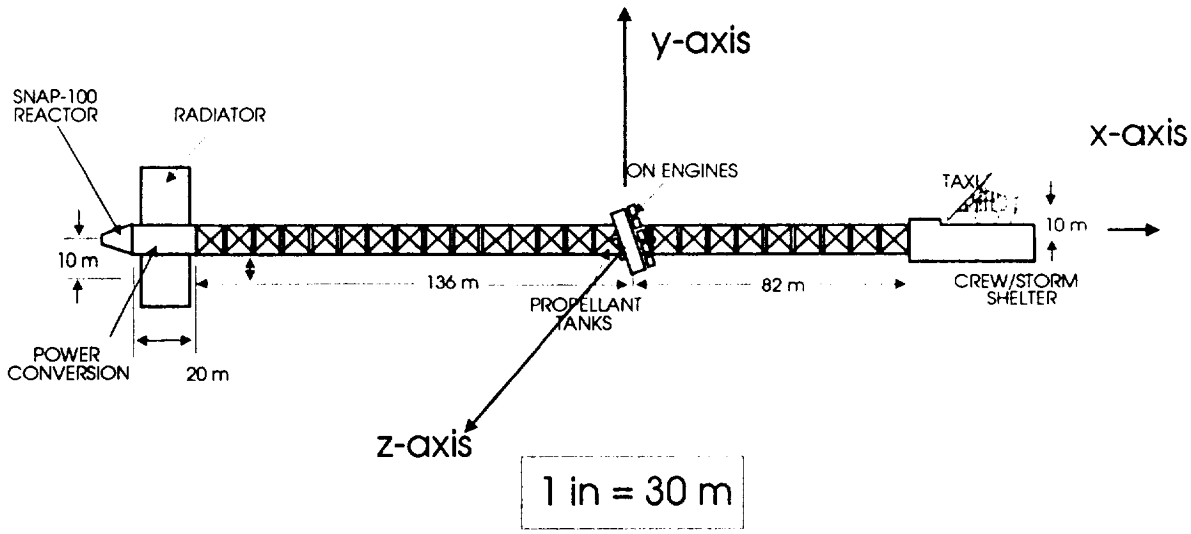


Figure 2.3c: A structurally simple cycler design. Ion engines located on the de-spun shelf.

Propulsion/Power Systems

In accord with the mission statement, the primary purpose of this project was to design a cycler spacecraft which utilized an ion propulsion system. The most important choice which had to be made by the design groups was deciding on the required level of thrust. The thrust level directly affected mass and size of the thruster array, the amount of power the vehicle electrical system needed to supply, and the length of time required to establish the cycling orbit upon launch from its assembly in the Earth-Moon system. To numerically estimate power and current requirements for the ion propulsion system the following equations were used

$$\text{Thrust} = \dot{m}\, g_0\, \text{Isp}$$
$$\text{Voltage} = (m_i\, \text{Isp}^2\, g_0^2)/(2e)$$
$$\text{Power} = (\text{Isp}\, g_0\, \text{Thrust})/(2\eta)$$

where g_0 = 9.81m/s^2, $\dot{m}$ = mass flow rate, m_i = mass of argon ion, e = elementary charge, η = efficiency of conversion. One notes that power is a linear function of Isp and thrust. For a 100 N thruster the required power is 5.45 MW. Needless to say this is a rather large requirement for a space borne nuclear reactor. Because the Cycler spacecraft typically has a mass upwards of 600,000 kg, fully fueled thrusters in the 30-100N range will be required for reasonable orbit insertion times. It is evident that the power requirements of the ion propulsion system will drive the requirements of the vehicle power system: For comparison, a chemically propelled cycler designed to carry 19 crew members and their associated life support and mission equipment is expected to require 300 kW of power[2]; this would be only 5.5% of the power required by a 100 N ion propulsion system. From these numbers it is apparent that a large part of the feasibility of developing an ion powered cycler spacecraft rests on the ability to construct and launch a nuclear reactor or alternative power generation system that can provide these large amounts of electric power. Over the 20-30 months period required to establish a final VISIT type orbit 52,000 to 78,000 kg of argon fuel would be required. Over the 40-50 months period expected to reach Up/Down Escalator orbits, the requirement is for 104,000 to 130,000 kg of fuel. These latter numbers are dependent on how optimum a thrust angle is achieved during powered flight and are thus subject to significant variation.

Table 3: Sample mass estimates. Numbers correspond to cycler and taxi spacecraft depicted in Figure 2.a.

Component	kg
Cycler	
Crew Compartment	25000 × 2
Reactor	4518
Habitant Consumables	2500
Structure	45000
Solar Arrays	30000
Argon Tanks	1200
LOX Tank	1500
H_2 Tanks	1200
Ion Thrusters	7500
Taxi	64638 No-Fuel
Total	212856
Taxi	
Crew Compartment	16700
LOX Tanks	1500 × 2
H_2 Tanks	1200 × 4
Cargo	17100
Thrusters	100 × 4
Landing Gear	1000 × 4
Structure	1500
Heat Shield	2000
Aero Shield	15138 No Fuel
Total	64638

It is unclear as to whether or not the ion engines are capable of providing the impulse required for the 3 mid-course adjustments of the Up/Down escalator orbits. To provide the largest single ΔV of 1.1 km/s the 100 N ion thrusters would need to operate for 76.4 days (assuming a spacecraft mass of 600,000 kg). Note that this is 9.8% of the total average 2.143 yr. period of these orbits. This point is significant to the extent that a secondary propulsion system might be required once the cycler has reached its orbit; this could be provided by a shuttle craft docked to the cycler or by an independent system on board the cycler. If deemed useful only for the initial orbital insertion, it might prove advantages to remove the ion propulsion system and/or the nuclear power generator after final orbital insertion is completed. The latter point is especially applicable for cycler spacecraft occupying VISIT orbits. An alternative, less massive, and more reliable power source such as solar panels could then be deployed to supply the roughly 300 kW of power required by the remaining vehicle systems, effectively reducing the mass of the vehicle, thereby saving fuel in the case of the Up/Down Escalator cycler. Assuming 300 kW are required at all points of a vehicle's orbit and 17% efficiency, a gallium-arsenide solar array spanning roughly 10000 m^2 would be required to satisfy the power requirement at the aphelion of the Up/Down escalator orbits from beginning to end of life. Near Earth, large amounts of un-needed energy would therefore have to be dissipated. If the radiators used for the nuclear generator are still available, this should not prove to be a large constraint, although some of the mass savings would then be lost.

Thermal Systems

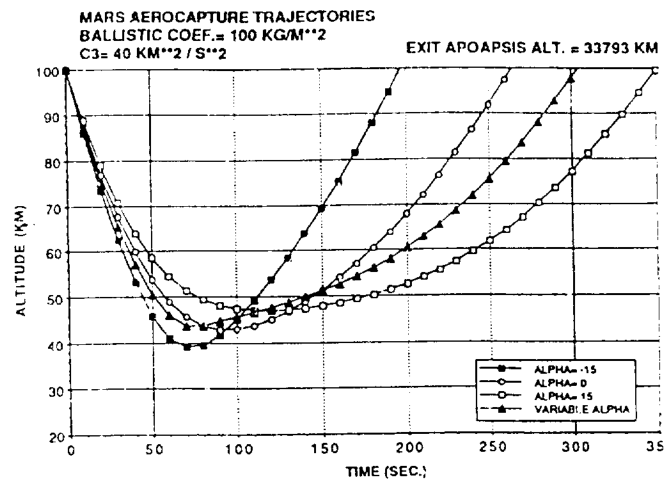
The primary thermal aspects of the project were centered around analyzing the boil-off problems of the cryogenic propellants and designing thermal blankets and/or refrigeration systems. Also included were the design ofthe radiators required to dissipate the heat generated by the power system, and the analysis of the heat loads caused during the aerocapture maneuvers at Earth and Mars.

The heat flux entering the cryogenic propellant tanks is primarily a function of solar radiation intensity and incident surface area, radiating surface area, and the thermal properties of storage tank and thermal blanket. The most efficient thermal blanket design seeks to maximize reflectivity and emissivity, and minimize absorptivity at the upper surface of each layer, while minimizing emissivity, and maximizing absorptivity on the lower surface of each layer. Except for the outermost layer which is exposed to optical wavelength radiation, the lower layers should achieve these values in the thermal wavelengths. Worst case design conditions occur at closest approach to the Sun, which occurs when the cycler flies by the Earth. Computer models were used to analyze the performance of the blankets given the thermal properties of each surface of a N-layer blanket. When the space between the layers was modeled as perfectly non-conducting, values of essentially zero heat flux were attainable with proper material selection. It became evident that the primary weakness of these thermal blankets are the spacers between layers. Assuming a worst case solar intensity of 1399 W/m^2, a sample thermal blanket using silver-FEP teflon on the upper surface and chromium on the lower surface of each layer, with a total of 6 layers and surface area of 500 m^2, achieved a total heat flux of .36 watts when tissue glass spacers were used. For a two year period, this magnitude of heat flux would result in a total boil-off of 50 kg of hydrogen. Assuming that highly efficient thermal blankets of this nature are realistic, it was assumed that boiloff would not be a problem. However, in case these efficiencies are not achievable, low temperature cooling systems were investigated. Reference 5 presents a good example of ideas in this field. The cooling system presented by Klein and Jones operates at 20 K, is capable of removing 0.48 watts, and weighs 31 kg. Multiple coolers of this type could be employed to accommodate the refrigeration needs of the LH2, argon, and LOX propellant tanks carried aboard the Cycler.

The primary trade-off in radiator design centered around high radiation temperature and associated low relative conversion cycle efficiency and small relative radiator surface area versus low radiation temperatures and associated high relative efficiency and large radiator surface area. Radiator design played a key role in most power system designs due to the large output required (up to 5.75 MW of power). Rankine, Stirling, and thermionic power conversion cycles were some of the systems considered. Preliminary analysis seemed to indicate that Stirling cycle systems would be required to provide the megawatts of power needed with the restraint of keeping the radiator size reasonable[6]. An example power system supplying 5.75 MW utilizing the Stirling cycle, which operated with a radiation temperature of 1000 K and a Carnot efficiency of 25% required radiators covering a 575 m^2 area.

Aerobraking requirements were strongly constrained by thermal considerations in the regimes dealt with by the vehicles in the Mars transportation system[2,7,8,9]. Current state of the art thermal protection systems (TPS), such as the reusable surface insulation TPS used on the space shuttle made up of ceramic tiles coated with a reaction cured glass, can withstand temperatures up to 1645 C. For most of the envisioned entry vehicles, especially those with L/D values above 1, aerobraking heat loads, especially those associated with Up/Down Escalator orbits, are expected to overwhelm current TPS systems. Therefore aeroshells using ablative materials were considered.

Due to the complexity of the heating analysis required for aerobraking maneuvers for Taxi, OTV, and cargo transfer vehicles entering the Mars atmosphere and the Taxi vehicle entering the Earth atmosphere, most of the analysis was restricted to what information was available in the literature. Typical loads associated with aerobrake trajectories at Earth and Mars are shown in Figure 2.4. The primary constraint on entry velocity into the Martian atmosphere seemed to be the atmosphere's ability to decrease the vehicle velocity enough in a single pass to ensure aerocapture; a quoted figure for the maximum approach velocity of a incoming vehicle is 9.97 km/s.



ALPHA (DEG)	TARGET PERIAPSIS (KM)	EXIT APOAPSIS (KM)	ENTRY VELOCITY (M/S)	MIN ALT (KM)	MAX G-LOAD (g)	MAX HEAT RATE (W/CM**2)	MAX DYN PRESS (N/M**2)
-15.0	27.0874	33793.22	8229.39	39.15	6.93	30.52	6628.69
0.0	43.4275	33791.81	8230.13	42.96	4.41	25.42	4320.50
15.0	51.9733	33810.53	8230.51	46.90	2.93	21.08	2800.86
-/+ 15.0	36.6991	33793.68	8229.82	43.74	4.71	26.74	4500.52

Figure 2.4: Sample aerocapture trajectories comparing angle of attack effects on flight loads, trajectory parameters, and final orbital conditions[7].

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TRIPOD LANDING STRUCTURE

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Abstract

The students were asked to conceive and construct a model of a tripod landing structure for a spacecraft landing on a planet. This was an exercise in space hardware design that made the students familiar with 3-D truss analysis and the study of various structural failure modes.

Design

Each design team was composed of four members, with all teams given entire freedom in the design of their landing structure as long as the following design criteria were met: 1) The truss had to connect three strong points, (the supports for the body of the spacecraft) with three landing pads. In the model, the support points had toform an equilateral triangle with 36.6" sides. 2) A height of the model of 18.0" was prescribed. 3) A minimum 6.0" high tetrahedral exclusion zone was required to take into account rocks and other planetary debris that might be on a planet's surface. For the same reason, no horizontal members directly connecting the three foot-points were permitted. 4) All dimensions had to be maintained to within 1/16". 5) Total weight of the structure was not to exceed 7.5 lb (Figs. 3.1a-c).

With these design criteria in mind, each group set out to find the best design for their respective landing structure. Each design would be judged on the basis of two criteria: a) highest ratio of ultimate load to weight; and b) closest agreement between calculated and actual ultimate load.

In designing the landing structure, each group recognized that buckling of the compression members would be the governing failure mode. Therefore, each group developed methods of designing around this problem. Some of the solutions included additional supports from tension members to compression members, doubling and tripling compression members, employing a concentric sleeve to strengthen the compression members, and using combinations of the above ideas. In all, twelve unique tripod landing structure designs were arrived at. To analyze these increasingly complex designs, various structural analysis packages such as NASTRAN and PATRAN were available. However, to find the predicted ultimate loads, most students still pursued the traditional pencil and paper analysis.

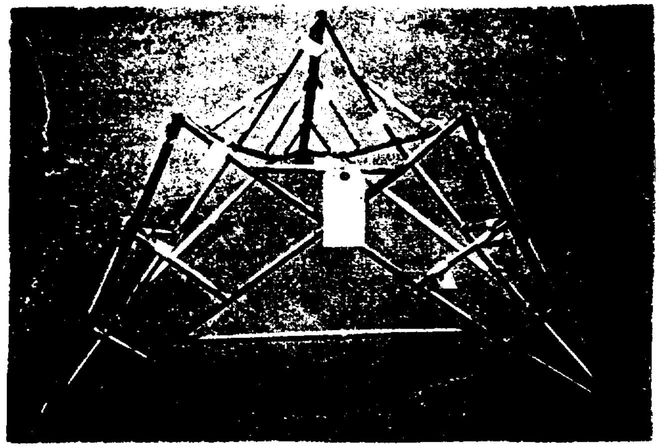

Figure 3.1a: Example of Euler Buckling.

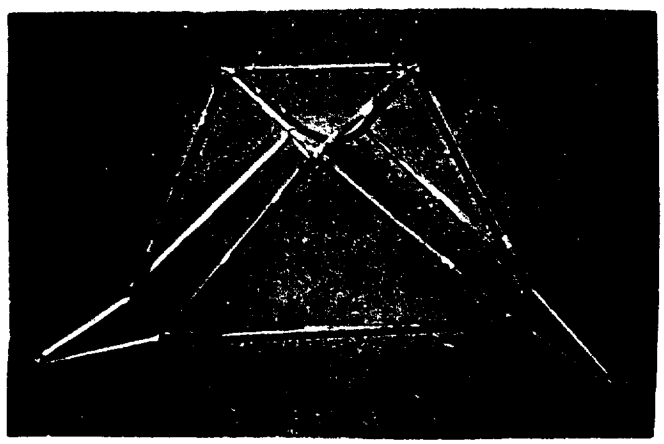

Figure 3.1b: Example of Bending in Compression Strut.

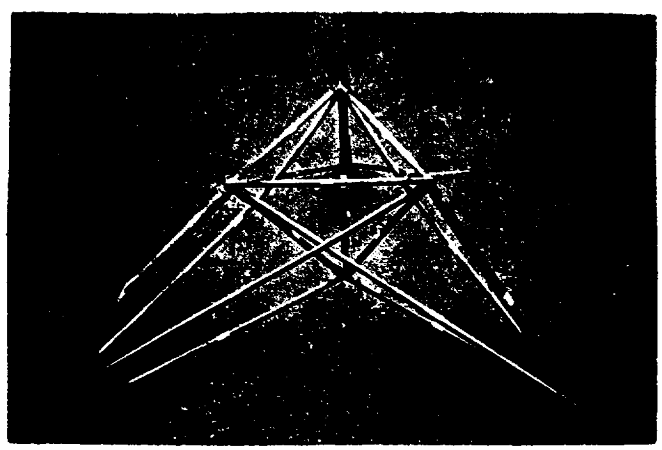

Figure 3.1c: Crack in Weld.

Testing

After the designs were completed, each group was responsible for the machining of their own truss structures from 1020 carbon steel tubing with a yield strength of 65,000 psi, ultimate strength of 80,000 psi, and Young's Modulus of 2.91×10^7 psi. This tubing was available in either 1/2" outer diameter at 0.1738 lb/ft., or 3/4" outer diameter at 0.2672 lb/ft., both with wall thicknesses of 0.035". With the tubes cut to be designed lengths and the proper intersection angles, the truss structures were put together by a professional welder.

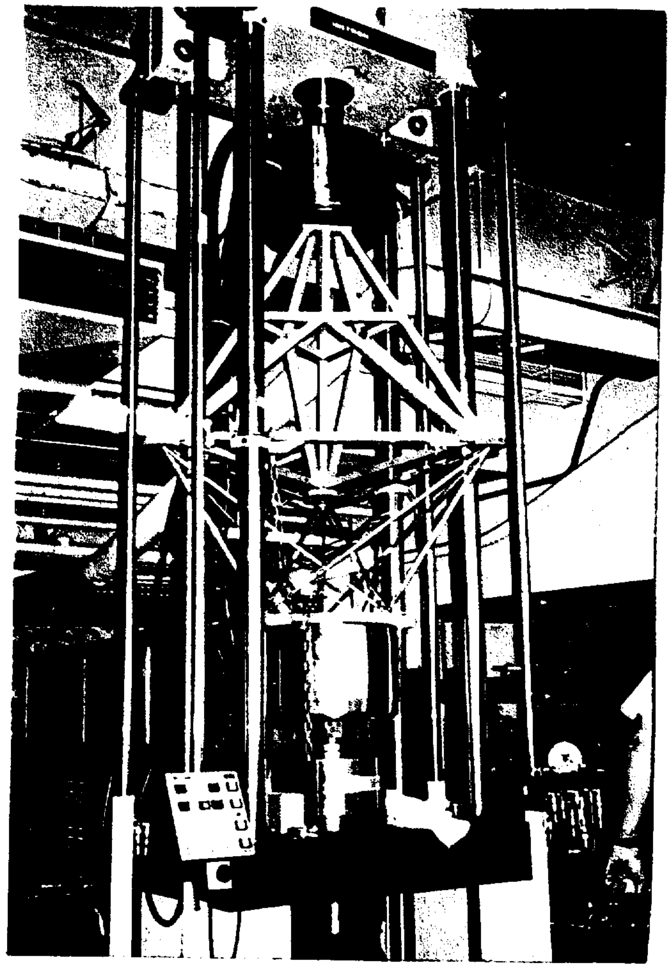

Figure 3.2: Tripod on Instron Test Machine.

To find the actual ultimate load, a 50 ton hydraulic press was utilized (Fig. 3.2). The results of the testing showed that the failures of the tripod landing structures fell into three basic categories: (a) Euler buckling; (b) thin wall buckling; and (c) cracked welds. One interesting failure mode that had occurred previously was lack of structural rigidity. In this failure mode, the structure has a tendency to twist or rotate when compressive loads are applied due to an insufficient number of bracing members.

The testing of the tripod landing structures resulted in some interesting observations. The truss with the highest load capacity 9105 lbs, (load-to-weight ratio of 1225) failed in tension, and therefore was successful in meeting the group's design goal of strengthening the compression members. Other groups were not as adept in this area. However, even though this group had achieved the highest maximum load, it did not have the highest load-to-weight ratio. This distinction was achieved by another group (load capacity of 8499 lbs, load-to-weight ratio of 1384).

The best agreement between measured and predicted maximum load was achieved by Group 8 with a ratio of 0.926. The predicted load closely approximated the actual maximum load because it was based on a relatively simple design that included no doubled, tripled, or concentric sleeve compression members.

Not only did the students gain exposure to both theory (truss analysis) and practice (minimizing the weight of the structure), but insight was gained into how to build optimal landing structures.

Table 3.1: Results of Tripod Testing

Group	Weight (lbs.)	Predicted Max. Load (lbs.)	Measured Max. Load (lbs.)	M.M.L./wt.	M.M.L./P.M.L.	Failure Mode
1	6.71	7217	6450	961	0.894	Crack in weld
2	6.34	8767	1650	260	0.188	Bend in comp. strut
3	7.44	7000	1200	161	0.171	Fold at hinge point
4	7.25	934	8720	1203	9.34	Bending at hinge point
5	6.83	5700	4391	643	0.77	Euler Buckling
6	7.43	4161	9105	1225	2.19	Tension membrane failed
7	7.29	3500	1491	205	0.426	Bending below joint
8	6.03	4137	3830	635	0.926	Euler buckling
9	6.14	7356	8499	1384	1.16	Thin wall buckling
10	6.99	3675	1024	146	0.279	Bending
11	7.18	5414	3786	527	0.699	Euler buckling
12	5.30	3260	4835	915	1.48	Euler buckling